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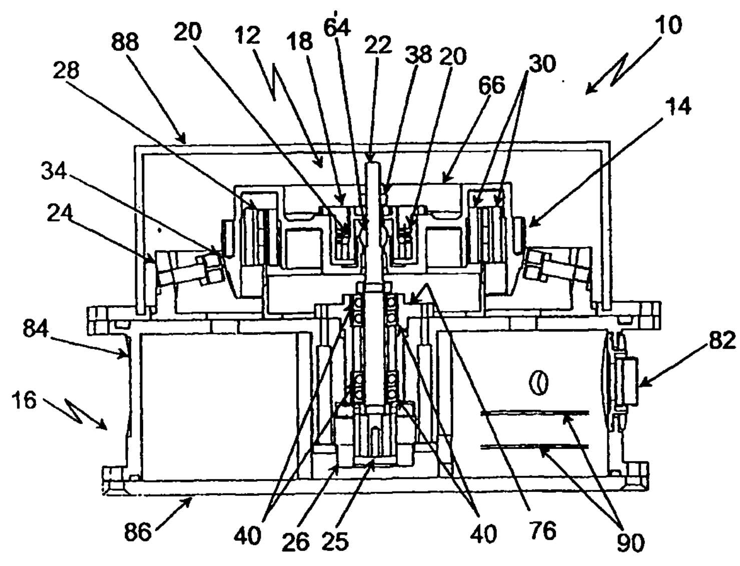

(57) Abstract

A momentum management system (10) for spacecraft attitude control. The momentum management system (10) includes a rotor (14) that provides control torques to a spacecraft. A drive (25) is provided to rotate the rotor (14), and a torque generation device (27) to impart torque to the rotor (14). A gimbal assembly (12) couples the drive (25) to the rotor (14). The gimbal assembly (12) has a spinning gimbal (18), and attaches to the drive (25) and the rotor (14) through the use of flexure joints (20). These flexure joints (20) permit the rotor (14) to tilt in two axes relative to a drive shaft (22), through a range of angles from about 0 degrees to about 7 degrees under the control of the torque generation device (27).

SYSTEM AND METHOD FOR SPACECRAFT ATTITUDE CONTROL

FIELD OF THE INVENTION

The present invention relates to spacecraft attitude control systems and methods. In particular, the present invention relates to a system and method for momentum management in spacecraft attitude control.

BACKGROUND OF THE INVENTION

Spacecraft attitude control systems, such as those used to control the attitude of satellites, are based on the direct control of angular momentum. The goal of such systems is to point a satellite, or portions of a satellite, at the earth, other celestial bodies, or another spacecraft. Generally, attitude control is achieved by maintaining a non-zero momentum state either by spinning the spacecraft body, or by including spinning bodies within the spacecraft. Spinning spacecraft include a separate despun platform on which the antennas and instrumentation are placed. Such spacecraft are termed dual spinners, or spinners. The alternative, where the main body of the spacecraft does not spin, and the angular momentum is provided by internal rotors, is generally called a body stabilized or three axis spacecraft. The present invention is intended for such three axis spacecraft, whether in geosynchronous or low earth orbit missions.

Geosynchronous satellites are primarily used for telecommunications. They are maintained in a near circular orbit above the equator at an altitude such that the orbit period is twenty four hours and the satellite remains nominally fixed in position with respect to a point on the Earth's equator. The orbit is not free from perturbations, and thruster forces must be applied periodically to maintain station. These station keeping manoeuvres either correct for orbit inclination disturbances, which effect north and south motions of the satellite, or for east and west accelerations. There are several distinct phases in the control of a geosynchronous satellite.

A highly elliptical transfer orbit is used to move a satellite from the Earth to injection into the final geosynchronous orbit, using the thrust from an apogee engine. This transfer orbit is often accomplished with the spacecraft rotating slowly at roughly 5 revolutions per minute. During the thrusting phases, and depending on the spacecraft inertial properties, a nutation or wobbling type of motion may develop unless direct control is exercised. This direct control may be accomplished by applying appropriate torques to the internal rotors under the direction of signals generated from gyroscopes, accelerometers or by other means. It is also

possible to remove the effects of an unbalanced spacecraft by applying appropriate torques to the internal rotors.

Once the final orbit is achieved, the spacecraft spin rate is reduced to zero and the normal pointing mode acquired. During this transition phase from a spinning state to a non-spinning state, a number of sensors may be used, including sun sensors, earth sensors, and gyroscopes, with control obtained directly by thrusters or indirectly by applying torques to the internal rotors. If the conditions are correct, it may be possible to transition directly from the transfer orbit spinning state to the non-spinning on-orbit state by means of an exchange of angular momentum between the internal rotors and the spacecraft body, without the need for external thruster sources of torque.

Most of the life of a spacecraft is spent quietly pointing at the earth or other target, with the control system absorbing external disturbances to maintain pointing. The major source of external disturbance is due to solar radiation pressure which creates a disturbing torque. Because this disturbance is very small, it is possible to provide a full three axes of control while actually measuring attitude pointing errors in two axes. The yaw axis joins the spacecraft with the earth's centre and rotations about the yaw axis are difficult to measure on a continuous basis. However, if the rotor angular momentum along an orthogonal axis is sufficiently large, the yaw rotation error can be constrained to remain within an acceptable envelope, through a form of gyrocompassing. The rotor size and angular momentum is determined by the allowed yaw errors and the disturbance torque through the well known relation:

$$\psi = \frac{T}{\omega_o h}$$

With ψ the yaw attitude error, T the disturbance torque, ω_o the orbit angular velocity and h the rotor angular momentum. The yaw attitude control accuracy can be improved, and/or the required rotor angular momentum can be reduced if angular velocities can be used to generate the control actuation in addition to the two axes of attitude pointing errors obtained from an Earth sensor, or other sensor source.

Direct control of the other two axes, the roll and pitch axes, is accomplished by managing the internal angular momentum state. Thus, for example, by torquing between the spacecraft body and a pitch rotor, the rotor can be made to change speed in one direction and the spacecraft will change speed in the opposite direction. Similar control actuation can be obtained by simply tilting a spinning rotor in the direction of the desired torque. The increase in angular

momentum along the desired axis due to the component of the rotor angular momentum along this axis is equivalent to applying a control torque.

When the station keeping manoeuvres occur, often on a two week operational cycle, the thruster activities result in disturbance torques which are orders of magnitude larger than during normal operation. Thus, the yaw pointing error can no longer be constrained with the normal rotor angular momentum. As a result it is necessary to measure changes in the yaw pointing error and actively provide control. It is common to use a yaw gyroscope, for the duration of the manoeuvre, to measure the yaw errors. Control is usually exercised through modulation of the thrusters which are used for the station keeping operation, or special purpose attitude control thrusters will be used.

Because failures do occur, it is necessary that any attitude control system address failures as an intimate part of the design process. For example, angular momentum is conserved, so that even if speed control of the internal rotors should fail for any reason, momentum will be transferred to the spacecraft body. In this case the spin axis direction is known and spacecraft health is maintained even though the operation may be interrupted. Ideally, in the event of a failure, the system and its components are capable of reconfiguration to redundant units so that no operational outage occurs.

The low earth orbit satellite (LEO) missions are quite different from geosynchronous satellite missions. A LEO satellite is usually injected directly into its final orbit by the launch vehicle so there is no transfer orbit. The Earth's magnetic field is sufficiently intense that it can be used as both an attitude sensing source (like a magnetic compass) or as a source of actuation. In addition, the orbital angular velocity is very much larger than at geosynchronous orbit so that the yaw error can be constrained, under normal modes of operation, with a smaller rotor angular momentum. However, during orbit adjust manoeuvres, it is necessary to measure all three attitude angles just as in the case of the geosynchronous satellite.

There are only a limited number of ways that high accuracy three axis attitude control is currently obtained. All are based in some way on control of stored angular momentum. For high accuracy, it is possible to use three or more reaction wheels, so that the total angular momentum magnitude and direction can be controlled, within the spacecraft body, by varying the speeds of each of the wheels. The wheels are usually mounted in an orthogonal triad. To provide redundancy, an additional wheel must be supplied for each axis, a costly and heavy approach. Identical control versatility can be obtained with four wheels mounted in a skewed configuration so that any three wheels can be used for control in the event of the failure of any

one wheel. A major drawback of the multi-wheel configurations is the number of wheels, with attendant redundancy, and the associated duplicated electronic boxes which are needed.

An alternative momentum management system uses the double gimballed momentum wheel in which a single momentum wheel is mounted within a two axis gimbal system. Actuation of the gimbals provides control of the angular momentum within the spacecraft body as required for control while limiting the total number of rotors to just a prime and redundant system.

A conventional double gimballed wheel consists of a momentum wheel mounted on a platform which can articulate. A trio of stepper motor driven linear jack screws is used to provide the tilt capability. The three actuators are needed to provide some redundancy in each wheel since jack screws can wear and eventually fail. A total angular deviation of about 6 degrees is adequate for this configuration. In principal, the double gimballed wheel has all the advantages of three reaction wheels with a momentum control capability in all directions while using but a single wheel. However, it suffers from two major disadvantages. The first is a result of the actuators which operate in discrete steps. This limits the pointing accuracy and requires careful nutation control. The second disadvantage is the complex mechanical configuration. It has numerous points of possible failure. These extra mechanisms add mass and cost and unreliability which are not desirable for small, low cost satellites. It is noteworthy that all of the current momentum management approaches, provide momentum control only and cannot be used to measure body rates as well.

It is, therefore, desirable to provide a novel system and method for momentum management in spacecraft attitude control that obviates or mitigates the disadvantages of the prior art.

SUMMARY OF THE INVENTION

The present invention generally provides a momentum management system for spacecraft attitude control. The momentum management system includes a rotor that provides control torques to a spacecraft. A drive is provided to rotate the rotor. and a torque generation device to impart torque to the rotor. A gimbal assembly couples the drive to the rotor. The gimbal assembly has a spinning gimbal, and attaches to the drive and the rotor through the use of flexure joints. These flexure joints permit the rotor to tilt in two axes relative to a drive shaft, through a range of angles from about 0 degrees to about 7 degrees under the control of the torque generation device.

In a preferred embodiment, the momentum management system flexure joints are formed by a cross flexure pivot formed of two flexures at right angles to one another. The gimbal assembly includes a launch restraint system, such as deflection stops, to limit deflection of the rotor under heavy loads. The torque generation device is formed of permanent magnets mounted inside the rotor, and torque coils that provide an inductive field to the permanent magnets. The momentum management system of this preferred embodiment also includes sensors for measuring tilt and speed of the rotor at non-tuned speeds and non-zero angles. The tilt sensors sense the edges of a triangular pattern etched on an outer surface of the rotor.

In a further aspect, the present invention provides a gimbal assembly for a momentum management system. The assembly includes a gimbal ring and flexure joints for coupling a drive shaft to a rotor. The flexure joints permit the rotor to tilt in two axes relative to the drive shaft through a range of angles from about 0 degrees to about 7 degrees under the control of a torque generation device.

In still another aspect of the present invention, there is provided a method of manufacturing the gimbal assembly. The method comprises the steps of machining an inner ring for insertion in an outer ring; mounting the inner and outer rings on a specialized jig; machining flexures through lateral sides of the inner and outer rings; inverting one of the inner and outer rings to align the flexures perpendicular to one another; and welding the inner and outer rings to form a gimbal ring. In a presently preferred embodiment, the machining is done by wire electric discharge machining. The tuned speed of the momentum management system can be adjusted by adjusting the height of the finished gimbal ring.

In yet a further aspect, the present invention provides a spacecraft attitude control system. The attitude control system includes a momentum management system as described above and a control system that provides control signals to the momentum management system. The control system can include a WHECON controller and a rate feedback controller.

BRIEF DESCRIPTION OF THE DRAWINGS

Presently preferred embodiments of the present invention will now be described, by way of example only, with reference to the attached Figures, in which:

Figure 1 is partial cut-away view of the momentum management system according to the present invention;

Fig. 2 is a cross section of the system of Fig. 1;

Fig. 3 is an isometric view of a gimbal according to the present invention;

Figs. 4a and 4b are schematic views of a cross flexure pivot according to the present invention, showing the flexure pivot with no relative rotation and with relative rotation, respectively;

Figs 5a and 5b show isometric views of outer and inner gimbal rings, respectively;

Fig. 6 shows a perspective view of a gimbal manufacturing jig according to an aspect of the present invention;

Figs 7a and 7b are cross sectional views of a first embodiment of a launch restraint system according to the present invention, showing the rotor in an undeflected and a deflected position, respectively;

Fig. 8 shows a cross section of a flexure pivot implementing a second embodiment of a launch restraint system according to the present invention;

Fig. 9 is a cross section of torque coils and axial rotor magnets according to one aspect of the present invention;

Fig. 10 is a cross section of torque coils and radial rotor magnets according to a further aspect of the present invention;

Fig. 11 is a block diagram of a the electronics for the system of the present invention;

Fig. 12 is a block diagram of a motor speed control circuit of the electronics of Fig. 11; and

Fig. 13 is block diagram of an attitude control system according to the present invention.

DETAILED DESCRIPTION

An embodiment of a momentum management system 10, according to the present invention, will now be described with reference to the attached Figures. Momentum management system 10 is a form of double gimballed momentum wheel based on a spinning gimbal assembly 12, as opposed to the conventional non-spinning gimbal. A spinning gimbal provides the capability to control the angular momentum in three axes, and to provide two axes of angular velocity measurement at the same time.

System 10 is shown in partial cut-away and in cross-section in Figs. 1 and 2, respectively. System 10 generally consists of a rotor 14 mounted in a housing 16, and suspended by gimbal assembly 12. Gimbal assembly 12 generally consists of gimbal ring 18, orthogonal flexure joints 20, a launch restraint system 38, a drive shaft 22 and bearings 40. Gimbal

assembly 12 behaves somewhat like a universal joint. If the spin speed is set to a singular value, system 10 will become tuned such that rotor 14 behaves as a free rotor, as though fully decoupled from drive shaft 22. With tilt sensors 24 that measure the tilt angles of rotor 14, and with a set of torque generation devices 27, the rotor tilt angles can be measured and controlled.

If the tilt angles are controlled to remain at zero, then the torque required to maintain rotor 14 at a its null position will provide a direct measure of the external angular velocities applied about the two axes lying in the rotor plane. This action is the basis of the much smaller, restricted motion, tuned rotor gyro which has found ready acceptance in the inertial navigation community and has demonstrated very long life capability in space versions.

System 10 provides the large angular momentum needed for spacecraft control and operates over an angular tilt range of ±7 degrees or more with speed variation of ±10 percent or more from the tuned speed. Angular tilt sensors 24 measure the rotor tilt angle and torque generators 30 provide the necessary torque to maintain a non-zero tilt angle. For system 10, the extra torques needed to overcome the non-resonant speed and non-zero tilt angle are highly predictable and can be modelled directly in terms of the wheel parameters and with calibration. Precise measurement of any excess or shortfall in the control torque will then provide a measure of the spacecraft body angular velocities.

Applications of system 10 to spacecraft control include: transfer orbit nutation control, attitude acquisition rate control, on-orbit momentum management, station keeping three axis control, and flexible body control.

Once the principals are apparent, it is possible to undertake a detailed analysis of the dynamic configuration. In its basic form, system 10 consists of three rigid bodies coupled by flexure joints. The three bodies are drive 25, consisting of a motor 26 and its drive shaft 22, gimbal ring 18, and rotor 14. Drive shaft 22 is coupled to gimbal ring 18 via a flexure 20a and gimbal ring 18 is in turn coupled to rotor 14 by another flexure 20b. A full non-linear three body analysis, as described in detail below, can be simplified considerably by assuming that the tilt angles are small in which case linear equations result and the problem reduces to an equivalent two body problem in which motor 26 is controlled to a constant speed. The linear analysis provides insight but is not completely accurate as the tilt angles increase beyond more than a few degrees. A full non-linear analysis of the three body problem cannot be solved explicitly but can be simulated. In this case it is found that if the motor speed control system is derived from error signals obtained from rotor 14 rather than from motor 26, and with a wide bandwidth, then the

three body system approaches the two body linear system in performance. These results allow for calibration of system 10 as a rate sensor as well as an actuator.

Spinning rotor 14 provides the angular momentum and also has a number of other features that permit the operation of system 10. The spin rate of rotor 14 is controlled by motor 26. The position of rotor 14 is controlled by torque generation devices 27, consisting of permanent magnet rings 30 and torque coils 28. Two permanent magnet rings 30 are mounted inside rotor 14 as shown in Figs. 1 and 2. Permanent magnets 30 are used to provide a means to impart torques on rotor 14 using torque coils 28 as described below. Rotor 14 also has a triangular pattern 32 etched into ferromagnetic material on the outer edge of rotor 14 as shown in Fig. 1. This etched pattern 32 is used with inductive pick-offs 34 in tilt sensors 24 to establish the tilt angles of rotor 14. The tilt sensor operation is discussed below. Finally, the top surface of rotor 14 is polished to a mirror finish to allow for fine calibration of system 10. With the permanent magnet design shown in Figs. 1 and 2, the rotor material is non-ferromagnetic (i.e., stainless steel), except for rotor pattern 32 which is machined into a ferromagnetic ring 36 press fit on stainless steel rotor 14. Permanent magnets 30 and the corresponding torque coil configuration is further explained below.

Gimbal 18 attaches rotor 14 to drive shaft 22 and uses two pairs of cross-flexure pivots 42 that permit rotor 14 to tilt about two axes orthogonal to shaft 22. An isometric view of gimbal 18 is given in Fig. 3. Cross flexure pivots 20a and 20b use two thin beam like strips of material, flexures 42a and 42b, are arranged in a cross configuration shown schematically in Figs. 4a and 4b. The cross configuration allows the top face to rotate relative to the bottom face while keeping the centre of rotation very close to the intersection of the undeformed strips, called the pivot centre 44. The z-axis pivots in Fig. 3 connect drive shaft mounting flange 45 to gimbal ring 18, while the x-axis pivots connect gimbal ring 18 to rotor mounting flange 46. The material used for flexures 42 needs to have a high fatigue limit so that the stresses in flexure 42 during normal operation remains well below this stress value to ensure essentially infinite life of flexure 42. In a presently preferred embodiment, the material used for gimbal 18 and flexures 42 is AerMet 100 which has a yield stress of nearly 300 ksi and a fatigue limit of over 100 ksi. Gimbal 18 also has provisions for balance screws 47 shown in Fig. 3 that can be adjusted during calibration to ensure the mass centre of gimbal ring 18 is located at pivot centre 44.

When spinning rotor 14 is oriented so that there is an angle between drive shaft 22 and spin axis, gimbal ring 18 tends to flutter with a frequency of twice the spin frequency. The tuned speed of system 10 is that which causes the inertial forces resulting from the gimbal

ring flutter to counter the torsional spring forces arising from deformed flexures 42 so that, to a first order approximation, rotor 14 behaves very nearly like a free rotor in space. This is a feature that permits system 10 to be used as a precise rate sensor. Gimbal 18 is designed so that the tuned speed of system 10 can be adjusted by simply changing the height of gimbal ring 18 without altering any of the interfaces between gimbal 18 and rotor 14 or drive shaft 22. Fig. 3 shows the full height gimbal ring 18 which corresponds to the lowest possible tuned speed. A reduced height gimbal ring corresponds to a higher tuned speed. This easily adjustable tuned speed permits system 10 to be easily adapted for a variety of speed ranges, making it applicable to a broad class of spacecraft. The reduction in gimbal ring height can be easily implemented for each individual system 10 produced, thereby tailoring the tuned speed to specific customer requirements.

Finally, drive shaft mounting flange 45 and rotor mounting flange 46 are designed to deform under launch loads in addition to flexures 42 to significantly increase the deflection of rotor 14 relative to drive shaft 22. The magnitude of the rotor deflection under load should be sufficient to implement launch restraint system 38 described below.

The gimbal design uses two concentric cylinders 48, 50 that are arranged so that flexures 42 can be easily machined into the cylinder blanks with both inner and outer flexures 42a and 42b being initially aligned. The two separate cylinders are shown in Figs. 5a and 5b, respectively, with flexures 42 machined (although not yet released). Inner cylinder 50 is then rotated 180 deg. relative to outer cylinder 48 to form the cross-flexure configuration and two cylinders 48, 50 are welded at the top and bottom. This process permits very cost effective manufacturing and also minimizes the effects of any manufacturing errors.

To implement this procedure a manufacturing jig 56 is used, as shown in Fig. 6. The manufacturing process commences when inner and outer cylinder blanks 48, 50 are first machined separately. They are then assembled and mounted into gimbal manufacturing jig 56. Alignment pins 58,59 on jig 56 are placed into corresponding alignment holes 60, 62 on rotor and shaft mounting flanges 45, 46, shown in Figs. 5a and 5b. Inner and outer gimbal rings 48, 50 are secured in jig 56 using attachment screws. Flexure strips 42 are then machined into the cylinder blanks, using, for example, a wire Electric Discharge Machining (EDM) process while the cylinders are in manufacturing jig 56. Flexures 42 are not yet released at this point. Outer cylinder 48 is then removed from jig 56, rotated 180 degrees and placed back into manufacturing jig 56. An alignment hole on the other side of rotor mounting flange 46 now engages with alignment pin 58 in jig 56. This procedure creates the cross-flexure configuration. Inner and

outer cylinders 48, 50 are welded together at the top of gimbal 18. Gimbal 18 is then removed and turned upside down and fastened to manufacturing jig 56 to permit the two cylinders to be welded at the bottom of gimbal 18 as well. Gimbal 18 is removed from jig 56 and turned around and fastened with the top face exposed (i.e., the side with rotor and shaft mounting flanges 45, 46). A sink EDM process can then used to remove the material required to release flexures 42 as shown in Figs. 5a and 5b by the dashed lines on gimbal inner and outer rings 48 and 50.

Launch restraint system 38 limits the deflection of rotor 14 during launch to ensure that flexures 42 in gimbal 18 are not over-stressed. In a first embodiment, launch restraint system 38a uses a caging ball 64 that is on drive shaft 22 shown in Fig. 2 , and a spherical cup 66 on rotor hub 68. Launch restraint system 38a is shown in greater detail in Figs. 7a and 7b. Under normal operational conditions, the rotor 14 is suspended by the cross-flexure pivots 20 so that the spherical cup 66 in the rotor hub 68 does not contact the caging ball 64. The centre of rotation of rotor 14 is located at the centre of caging ball 64, so that as rotor 14 tilts a gap 70 between ball 64 and spherical cup 66 is maintained. Therefore, during normal operational conditions, caging ball 64 does not come into contact with rotor 14 and hence does not affect the operation of system 10. However, when loads are applied to rotor 14 during launch and also during handling, rotor 14 will be deflected until spherical cup 66 contacts caging ball 64 as shown in Fig. 7b. Caging ball 64 therefore limits the maximum deflection of rotor 14 and thus limits the maximum stress in gimbal flexures 42. Although Fig. 7b shows a rotor deflection in the vertical direction, caging ball 64 can limit deflection of rotor 14 in any arbitrary direction.

To obtain a practical gap 70 between rotor spherical cup 66 and caging ball 64, gimbal flexures 42 are sized to provide for the necessary deflection in both vertical and lateral directions. Also, rotor mounting flanges 46 are designed to also deform under load to increase the allowable gap.

Finally, to limit the angular deflection of rotor 14, stops 71 can be incorporated into rotor hub 68, such that as rotor 14 is tilted to the maximum allowable angle, rotor hub 68 contacts drive shaft 22. This provides a rigid stop for the angular deflection that limits the maximum stress in flexures 42 due to the tilt of rotor 14. Gap size 70 is selected so that flexures 42 cannot be overstressed even when rotor 14 is tilted to the maximum angle.

A presently preferred embodiment, that reduces machining and assembly, of the launch restraint system 38, is shown in Fig. 8. The launch restraint system 38b of this embodiment introduces deflection stops 72 that ensure that flexures 42 in gimbal 18 will not be overstressed. Deflection stops 72 can be machined out of the parent gimbal during the wire

EDM operation, described above, used to form flexures 42. A small gap 73, in the range of 0.1524 - 0.1778 mm, is formed between flexure 42 and deflection stop 72, such that under normal operating conditions of 1-g, flexure 42 does not touch deflection stop 72. However, under high loads, such as launch and handling loads, the lateral deflection of flexure 42 is restrained at the midpoint, allowing it to carry more load. The buckling behaviour of flexure 42 in this case is forced into a second mode shape that effectively increases the buckling load by a factor of 4. Hence, it is possible to design flexures 42 that will not be overstressed in the worst loading conditions.

Drive shaft 22 and bearings 40 are shown in Fig. 2. Spin motor 26 attaches to the end of drive shaft 22 and spins shaft 22 at the required speed. Shaft 22 connects to gimbal 18 at the top end and shaft 22 engages into mounting flange 45 on gimbal 18 to align gimbal 18 in a repeatable direction relative to shaft 22. Bearings 40 support drive shaft 22 between spin motor 26 and gimbal 18. Two pairs of R6 duplex bearings are used. Bearings 40 are housed in a separate thermal sleeve 76 made from the same materials as the bearings 40 and shaft 22 reducing any thermally induced stresses on the bearings 40. Separate thermal sleeve 76 also permits removal of the complete gimbal assembly 12 (i.e., shaft, bearings, gimbal and rotor) from system 10 without disassembling gimbal assembly 12. Drive shaft 22, bearings 40 and thermal sleeve 76 are a single subassembly that can be assembled and tested separately before installing into system 10.

Four (4) torque coils 28 arranged in two pairs are used to impart torques on rotor 14 about the x and z axes. Coils 28 are mounted to a torque coil stand 78, as shown in the Fig. 1, and are located between the two permanent magnet rings 30 in rotor 14. Coils 28 are made using a very fine copper wire (e.g., 32 gauge) wound around a form with a large number of turns (e.g., 400 turns). When a voltage is applied to coils 28, the resulting current in coils 28 interacts with the magnetic flux $\boldsymbol{B}$ in the gap between rotor magnets 30 imparting an axial force on rotor 14. A pair of coils 28, opposite each other, and with the current running in opposite directions, is used to impart a moment on rotor 14 without imparting a net force. The torque required to move rotor 14 or hold it fixed is related to the angular rates of the spacecraft. Hence, the measured current in coils 28 can be used to provide the rate measurements about the x and z axes.

A cross section of coils 28 and rotor magnets 30 is given in Fig. 9 showing the magnetic configuration and the direction of the magnetic flux in the gap where coils 28 are located. The configuration shown in Fig. 9 uses magnets that are axially magnetized. In this

case, the rotor material used is non-ferromagnetic allowing the flux to be concentrated in the gap between the magnets. Also with this configuration, the outer magnet width can be sized so that the magnetic flux external to the rotor can be made very small. Another possible magnetic configuration is one that uses permanent magnet rings that are radially magnetized. This configuration is shown in Fig. 10. In this case, rotor 14 is made from a ferromagnetic material to provide a flux path between the magnets.

In the permanent magnet design of Fig. 9, the axial length of the magnets relative to the torque coil geometry determines performance. If one plots the magnetic strength along the axial direction in the middle of the gap in between magnets 30, the flux will have sharp peaks at the ends of magnets 30. However, over the tilt range of rotor 14 it is highly desirable to maintain a constant torque scale factor (i.e., N-m/Amp) from torque coils 28 to improve the calibration. This can be accomplished by selecting the length of magnets 30 appropriately. If the magnets are too short, the torque scale factor is reduced as the tilt angle is increased. Similarly if the magnets too long, the torque scale factor increases as the tilt angle is increased. A similar situation exists with the radial magnets shown in Fig. 10, however in this case, the axial spacing between the upper and lower magnets is the determinant parameter. The radial magnets can have an advantage over the axial magnets as they tends to flatten out and widen the magnetic flux peaks which helps to further reduce the variation of the torque scale factor over the range of tilt angles.

Tilt sensors 24 consist of inductive pick-offs 34 that use a small sensing coil located very near machined pattern 32 on the outside spherical ferromagnetic surface of rotor 14, as shown in Fig. 1. Pattern 32 on rotor 14 introduces a changing gap size between tilt sensor 24 and the ferromagnetic rotor surface as rotor 14 rotates past one of tilt sensors 24. This changing gap introduces a signal in the tilt sensor coil 34 that can be used to measure the rotor tilt. The principle is that as a single triangular etched pattern 32 rotates past the sensors 24, the leading edge of the pattern 32 causes a current spike in sensing coil 34 and the trailing edge of pattern 32 causes a negative current spike. A typical waveform is shown in Fig. 11. Since pattern 32 is triangular, the time between the leading edge pulse and the trailing edge pulse is related to the rotor tilt for a given rotor speed. When tilt sensor 24 is near the top part of pattern 32, the pulses are further apart, and when sensor 24 is near the bottom part of pattern 32, the pulses are closer together. To remove the dependence on the rotor spin rate, the ratio T1/T2 is defined where T1 is the time between the leading edge pulse and the trailing edge pulse and T2 is the time between the leading edge pulses of two consecutive patterns. Then the tilt angle is proportional only to

the ratio T1/T2 (assuming the spin rate remains constant over the measurement interval). Tilt sensors 24 are arranged in pairs as shown in Fig. 1 where one pair is used to measure the tilt about the x-axis and the other pair is used to measure the tilt about the z-axis.

Spin motor 26 is preferably a brushless DC electric motor that is integrated into a base 82 in housing 16, as shown in Fig. 2. Permanent magnets are mounted on the shaft and the stator is integrated into housing 16.

Housing 16 consists the a case 84, a bottom cover 86, and a top cover 88. Top and bottom covers 86, 88 have O-ring seals and connectors that are hermetically sealed so that system 10 can be evacuated and placed into a vacuum. This reduces windage effects on rotor 14 and creates a more similar thermal environment to the in-space configuration to allow for more precise calibration on the ground.

Housing 16 is also designed to separate electronics 89 and spin motor 26, which are located in the lower compartment 82, from gimbal assembly 12 located in the upper compartment 87. This permits thermal isolation and active control of the rotor temperature while the lower compartment 82, where the primary power/heat dissipating devices are located, does not require active thermal control and can have a good thermal interface to the spacecraft.

A theoretical analysis of system 10 will now be provided, with reference to the preceding structural description. For present purposes it can be assumed that housing 16 and motor shaft 22 are fixed in inertial space. The three elements are motor drive shaft 22, gimbal ring 18 and rotor 14 itself. Motor 26 is driven at a constant speed ω_s. Gimbal 18 is connected to shaft 22 by a pair of torsion flexures 42a aligned along the x-axis. Rotor 14 is connected to gimbal 18 by another pair of torsion springs 42b aligned along the y-axis. The x and y axes are spinning around the spin axis (z-axis) and gimbal 18 can rotate about the x-axis while rotor 14 can in turn rotate about the gimbal's y-axis. In the following, the spinning coordinates are retained until the results are transformed to non-spinning coordinates appropriate for the angle sensors, torque generators (torquers) and the rebalance control loop.

Let φ be the angle of rotation of gimbal ring 18 about the x-axis and let θ be the angle of rotation of rotor 14 with respect to gimbal 18, about the y-axis. The two body dynamic equations may be derived using a vectorial or a Lagrangian dynamic formulation assuming that motor 26 speed is a constant. It will be assumed that the moment of inertia of gimbal 18 about the z or spin axis is I_{gs} and that gimbal 18 is symmetric with transverse inertia I_{gt}. Similarly the spin axis inertia of the rotor 14 is I_{rs} and the symmetric transverse inertia is I_{rt}. The dynamic equations that result are:

$$
\begin{aligned}
&(I_{rt} + I_{gt})\dot{\omega}_{rx}C_\theta - I_{rt}(\omega_{rx}\omega_{gy}S_\theta + \omega_{ry}\omega_{gz}) \\
&+ I_{rs}\omega_{rz}\omega_{gy}C_\theta + (I_{rs} + I_{gt})\dot{\omega}_{rz}S_\theta \\
&+ (I_{gt}\omega_{gz} + I_{rs}\omega_{rz}C_\theta - I_{rt}\omega_{rx}S_\theta)(\omega_{ry} - \omega_s S_\theta) \\
&- (I_{gt} - I_{gs})\omega_{gy}\omega_{gz} + k_x\phi = 0
\end{aligned}
$$

and

$$I_{rt}\dot{\omega}_{ry} + I_{rt}\omega_{rx}\omega_{rz} - I_{rz}\omega_{rz}\omega_{rx} + k_x\theta = 0$$

where:

I_{rs}	rotor inertia about the spin axis
I_{rt}	rotor inertia about the transverse axis
I_{gs}	gimbal inertia about the spin axis
I_{gt}	gimbal inertia about the transverse axis
k_x	x-axis flexure spring constant
k_y	y-axis flexure spring constant
φ	angle of rotation of the gimbal about the x-axis
θ	angle of rotation of the gimbal about the y-axis
C_θ	$\cos\theta$
S_θ	$\sin\theta$
C_φ	$\cos\varphi$
S_φ	$\sin\varphi$
ω_s	motor spin speed
ω_{gx}	gimbal angular velocity about the x-axis
ω_{gy}	gimbal angular velocity about the y-axis
ω_{gz}	gimbal angular velocity about the z-axis
ω_{rx}	rotor angular velocity about the x-axis
ω_{ry}	rotor angular velocity about the y-axis
ω_{rz}	rotor angular velocity about the z-axis

Using small angle approximations, these reduce to

$$(I_{rt} + I_{gt})\dot{\omega}_{rx} + (I_{gt} + I_{rs} - I_{rt})\omega_s\omega_{ry}$$
$$+\left[k_x - 2\left(I_{gt} - \frac{I_{gs}}{2}\right)\omega_s^2\right]\phi = 0$$

and

$$I_{rt}\dot{\omega}_{ry} - (I_{rs} - I_{rt})\omega_s\omega_{rx} + k_y\theta = 0$$

Define

$$J = \left(I_{gt} - \frac{I_{gs}}{2}\right)$$

and the nutation frequency

$$\omega_n = \frac{\left(I_{rs} + I_{gt}\right)}{\left(I_{rt} + \frac{I_{gt}}{2}\right)}\omega_s$$

and introduce the Laplace transform with the auxiliary variables:

$$\alpha = \frac{k_x + k_y - 2J\omega_s^2}{2I_{rt} + I_{gt}}$$
$$\beta = \frac{k_x - k_y - 2J\omega_s^2}{2I_{rt} + I_{gt}}$$
$$\gamma = \frac{I_{gt}}{2I_{rt} + I_{gt}}$$

the characteristic equation may be found in the form:

$$\left(s^2 + \omega_s^2\right)^2 + \frac{\omega_n^2 - 2\omega_n\omega_s + 2(\alpha - \gamma\beta)}{1 - \gamma^2}\left(s^2 + \omega_s^2\right)$$
$$+ \frac{2\alpha\omega_s(\omega_n - 2\omega_s) + \alpha^2 - \beta^2}{1 - \gamma^2} = 0$$

There are two solutions for $\left(s^2 + \omega_s^2\right)$

$$\left(s^2 + \omega_s^2\right) = \frac{\omega_n(\omega_n - 2\omega)_s + 2(\alpha - \gamma\beta)}{2(1 - \gamma^2)}$$
$$\pm \frac{1}{2}\sqrt{\left[\frac{\omega_n(\omega_n - 2\omega)_s + 2(\alpha - \gamma\beta)}{2(1 - \gamma^2)}\right]^2 - 4\left[\frac{2\alpha\omega_s(\omega_n - 2\omega_s) + \alpha^2 - \beta^2}{(1 - \gamma^2)}\right]}$$

There are also two more solutions when the square root is taken. Thus there are four roots in all. The classic condition for tuning a tuned rotor gyro is the condition that $\alpha = 0$, that is:

$$k_x + k_y = 2J\omega_s^2$$

and if $k_x = k_y = k$ say, then the classic tuned condition is $k = J\omega_s^2$.

The resulting roots may be transformed to non-spinning coordinates by adding ω_s and it will then be found that the first of the roots is a resonance at twice the spin frequency and the second root is the nutation frequency. The third root is an additional resonance which presumably represents a beat frequency between the gimbal oscillations and the rotor nutation. The last of the roots is at a very low frequency and will appear as a drift rate or slow precession as the result of an off-null rotor, precessing in a coning motion. This last root, which in the two body linear model can be precisely defined, provides a determination of the effect of spin speed and rotor tilt angle on control torque requirements. Consequently, since the roots may be calculated and calibrated accurately, the effect of a rotor tilt angle and non-tuned speed can be

precisely calculated and subtracted from the torquer measurements so that angular velocities may be measured under any rotor condition.

A higher degree of tuning can be obtained directly from the characteristic equation if the last term is set to zero. The resulting low frequency root will then be identically zero and no drift will be experienced. This more precise tuning condition is:

$$2\alpha\omega_s(\omega_n - 2\omega_s) + \alpha^2 - \beta^2 = 0$$

which may be approximated for small values of J to give:

$$k \approx J\omega_s^2\left[1 + \frac{J\omega_s}{(2I_{rt} + I_{gt})(\omega_n - 2\omega_s)}\right]$$

This may be considered as a condition on the flexure stiffness k. or on the spin speed ω_s. For system10, this precise tuned condition identifies the nominal spin speed. A more complete analysis can also be done by a three body analysis which includes the inertial properties of motor 26 and drive shaft 22, however, such analysis is beyond the scope of this disclosure.

A control system 90 is intended to provide high efficiency drive for motor 26; provide precise rotor speed control; process pulses for tilt sensors 24; provide drive of rotor torque coils 28; measure the torque current to the highest possible precision; monitor the system temperature; provide a microcomputer for general processing; provide a serial interface to the spacecraft attitude control system, or flight computer, to accept momentum commands (speed and 2-axis tilt), and possible software uploads; and support calibration of system 10 for temperature and non-linear effects.

A limitation with inertial measurement systems is often the electronics rather than the mechanical components. Control system 90 uses digital control to achieve the level of performance required, and includes electronics 89. Electronics 89 are built into the lower section 82 of housing 16 to control torque coils 28. Referring to Fig. 11, electronics 89 include a microcomputer 91 provided with appropriate software applications, A/D and D/A converters 92, 93, respectively, a phase locked loop 94, power amplifiers 95, and power supplies 96. While there are actually four torque coils 28, two drivers 97 have been found sufficient if pairs are connected in parallel or series. Four drivers 97 can be used if it is desired to provide soft failure redundancy. Electronics 89 further include a speed control 98, and a motor drive 99.

A block diagram of motor speed control circuit 98 is shown in Fig. 12. The required speed range for rotor 14 is typically 1300 - 1600 revolutions per minute (rpm). The speed is measured using a tachometer 102 with K_z pulses per revolution. The reference frequency for the phase locked loop 94 is derived using a ÷N counter 104. This gives non-linear control, but computer 91 can calculate the required N and the hardware is simpler that using an accumulating rate multiplier or DDS to synthesize the reference. The phase locked loop 94 uses a phase/frequency detector 106 to automatically acquire lock after a step speed change. The rotor 14 speed can be given, in revolutions per second (rps), by:

$$Speed = \frac{F_{ref}}{NK_z}$$

Where,

F_{ref} is the crystal oscillator frequency

N is the divider value

K_z is the tachometer pulses per revolution

Therefore, selecting K_z is a compromise between noise and speed resolution, both of which can affect spacecraft jitter. A large K_z reduces noise by increasing the frequency of the pulses at the phase detector. This shifts the frequency of the noise components at the output of the phase detector outside the bandwidth of the phase locked loop 94 where it can be attenuated by filtering. It also reduces the required gain after the phase detector which can reduce noise. However, a large K_z decreases the speed resolution, which increases jitter when the control system 98 switches speeds. Assuming a 10 MHz reference, $K_z = 8$, and 500:1 spacecraft to rotor momentum ratio, the spacecraft jitter caused by finite speed resolution can vary from 0.0004 to 0.005°/sec. Depending on the rotor speed.

Motor 26 consumes a significant portion of the power budget and a high efficiency motor drive 99 is required. The motor speed range is typically in the range of 1500 ±200 rpm to 5200 ±800 rpm. A suitable motor drive 99 should be ~ 90% efficient, and provide regenerative breaking when a negative torque is commanded.

Control system 90 also compensates for temperature dependent effects. The main source of temperature dependence is likely permanent magnets 30, particularly if ceramic magnets are used, which have a 0.6%/°C temperature coefficient. A thermistor sensor 110 can

provide temperature readings, and is unaffected by nuclear radiation. Microcomputer 91 processes the temperature readings to provide suitable calibration to system 10.

For earth pointing missions, the spacecraft pitch dynamics decouples from the roll-yaw dynamics which allows a pitch controller to be designed separately from a roll-yaw controller (note: the roll and yaw dynamics remain coupled due to the momentum bias). System 10 provides control torques about all three axes, and hence can be used for an attitude control system 150, as shown in Fig. 19. However, since it also provides rate measurements about the roll and yaw axes, this provides for significantly simplified and improved control system for the roll-yaw dynamics of the spacecraft. In particular, since the roll and yaw attitude errors are coupled due to the momentum bias, then system 10 is controllable by only using a roll measurement, which can be obtained using an earth sensor, and the roll and yaw rate measurements. Therefore, to achieve fine pointing control in all three axes with system 10, only a two-axis earth sensor is required that provides a pitch and roll angle measurement, where the pitch angle is used for the pitch control loop. This eliminates the need for a separate sensor (e.g., a sun sensor or a star camera) to directly measure yaw which simplifies the design of attitude control system 150 and significantly reduces the cost. Also, the performance of attitude control system 150 is not affected by going in and out of eclipse as it would be if a sun sensor is used to determine yaw as is the case for most earth pointing spacecraft.

A classical PD control system can be used for the pitch axis control where the control law is expressed as:

$$M_y = -K_p\alpha_y - K_d\dot{\alpha}_y$$

where K_p and K_d are the proportional and derivative gains respectively, α_y and $\dot{\alpha}_y$ are the pitch angle and pitch rate, and M_y is the control torque about the spacecraft pitch axis. The pitch angle α_y can be obtained from an earth sensor and the pitch rate $\dot{\alpha}_y$ is estimated using a finite difference scheme:

$$\dot{\alpha}_y = \frac{\alpha_y - \alpha_y^{pre}}{\Delta t}$$

where α_y^{pre} is the pitch angle at the previous sampling time and Δt is the controller time step.

For the roll-yaw control, a direct output feedback control structure can be used as follows

$$\begin{Bmatrix} M_x \\ M_z \end{Bmatrix} = [F]\begin{Bmatrix} \alpha_x \\ \omega_x \\ \omega_z \end{Bmatrix}$$

where M_x and M_z are the control torques about the spacecraft roll and yaw axes, [F] is the feedback gain matrix; α_x is the roll angle measurement, and ω_x and ω_z are the measured inertial rates of the spacecraft about roll and yaw axes respectively. This structure, as shown in Fig. 19, uses earth sensor measurements and measurements from system 10, directly and does not require an attitude determination algorithm to process the measurement data which further simplifies the ACS design. Since the roll and yaw axes are dynamically coupled, this control structure can provide fine pointing control in both the roll and yaw axes and there is no need to explicitly estimate the yaw angle (although this is also possible).

To establish appropriate values for the feedback gain matrix [F], a WHECON control structure 151 can be used with an added yaw rate feedback loop 152. The classical WHECON control algorithm makes use of the coupling between the roll and yaw dynamics of the system to express the control law as follows:

$$M_x = -K(\alpha_x + \tau\dot{\alpha}_x)\cos\phi$$
$$M_z = K(\alpha_x + \tau\dot{\alpha}_x)\sin\phi$$

where K is the proportional gain, τ is the damping gain, and φ is the constant offset angle (selected based on the spacecraft inertial properties), and α_x and $\dot{\alpha}_x$ are the roll angle and the roll rate. To improve the control performance particularly in yaw, an additional yaw rate feedback loop is designed using rate measurement about the yaw axis ω_z and also rate measurement about the roll axis ω_x is used in place of roll rate $\dot{\alpha}_x$, as provided by system 10. The roll-yaw control law therefore becomes as follows:

$$M_x = -K(\alpha_x + \tau\omega_x)\cos\phi$$
$$M_z = K(\alpha_x + \tau\omega_x)\sin\phi - K_r\omega_z$$

where K_r is the rate feedback gain ω_x and ω_z are the components of the spacecraft rates about the roll and yaw axes measured by system 10. The control law can be put into matrix form and the feedback gain matrix [F] then becomes:

$$[F] = \begin{bmatrix} -K\cos\phi & -K\tau\cos\phi & 0 \\ K\sin\phi & K\tau\sin\phi & -K_r \end{bmatrix}$$

This structure for the roll-yaw control law only has four parameters to be specified to establish the gain matrix $(K, \tau, \phi, \mathit{and}\, K_r)$, each of which has some practical interpretation for an orbiting, bias momentum spacecraft. Therefore, this makes it much easier to select an optimal set of gains that provide the best performance of attitude control system 150.

Another approach to establishing the gains is to specify the eigenvalues and eigenvectors for the closed loop roll-yaw dynamics. However, due to the coupling between the roll and yaw axes, it is difficult to relate the eigenvalues and the eigenvectors to the physical behaviour of system 10. Hence, it essentially becomes a trial and error process to some extent to select the appropriate parameters. Therefore, since one has to select 8 parameters for the closed-loop eigenvalues and 6 parameters for the eigenvectors to design the controller, it becomes a very difficult task to select the optimum set of parameters which will give the best performance for the system. The WHECON 151 with yaw rate feedback is therefore a superior controller structure that can provide very fine pointing performance in both roll and yaw axes.

The feedback control loop 152 for attitude control system 150 must provide both control of the tilt angle of rotor 14 as well as minimize the addition of drift errors for purposes of measuring rates. A successful feedback control loop 152 depends on the careful control of the feedback loop phase characteristics. The basic difficulties are inherent in any tuned rotor gyroscope rebalance loop and those skilled in the art of tuned rotor gyroscope rebalance loop design will be familiar with the requirements.

The basic purpose of feedback loop 152 is to provide precession control of rotor 14. A simple proportional-plus-integral controller is applied on each of the two tilt axes to ensure asymptotic tracking of the tilt demand even when the system is de-tuned. In addition, to account for the cross-axis control requirements for precession control, a -90 degree phase shift is introduced. The justification for this can be seen most easily by examining the complex variable formulation of the rotor dynamics. However, this simple and straightforward precession control system will excite the nutation dynamics of rotor 14. Thus, nutation damping must be

included. One method for handling the nutation dynamics, as represented in the complex variable formulation of rotor 14 dynamics, is to employ low pass filters that provides a full 180 degrees of phase shift at the nutation frequency. This ensures that as frequencies pass through the nutation resonance, the response locus with the added 180 degrees resonant phase lag and infinite gain will be restricted entirely to the right half plane, in a Nyquist analysis. It is necessary that the overall gain of this low pass phase lag loop be small enough so that the first unity gain cross over point occurs well before a 180 degree lag, that is, well below the nutation frequency. Consequently, this nutation damping approach assumes that the overall precession control bandwidth requirement is well below the nutation frequency.

Two other requirements are placed on the feedback control loop 152. Firstly, signals, which are inevitably generated by tilt sensor 24 at the spin speed of rotor 14, must be notched to reduce their amplitude to negligible size. Finally, the gimbal motions, at twice the spin frequency, introduce dynamic motions to rotor 14, and these will be sensed by tilt sensors 24. These twice spin frequency signals must also be notched to reduce their amplitude. However, it is not possible to reduce the amplitude of these signals precisely to zero and any portion that is fed back to the rotor controller will interact with the actual twice spin frequency dynamics. The phasing of the twice spin speed feedback signal can introduce an effective drift rate in the sensed external rates applied to the system 10. Thus, careful phase adjustment of the twice spin frequency notch filters is required to limit this drift effect. Care is needed in ensuring that the phase shifts for these notch filters and the nutation control lag filters continue to sum to roughly180 degrees, at the nutation frequency.

Some of the advantages and innovations of system 10, and attitude control system 150 can be summarized as follows. Use of a spinning mechanical suspension system for rotor 14 allows for momentum steering about two axes. Such an approach, where mechanical gimbal 18 with flexure pivots 20 is used to support spinning rotor 14 has never before been used for a torque actuator. There are other approaches that use a magnetic suspension system that permit two axis momentum steering, and mechanical systems that tilt a spinning wheel. The use of gimbal 18 with flexure pivots 20 for suspending rotor 14 has a number of advantages. It can be designed to have infinite life with no wear-out modes and hence has much higher reliability than the magnetic suspension systems. It allows for a very simple, compact and cost effective implementation approach using torque coils 28 and permanent magnets 30 in rotor 14 to effect the momentum steering. And, it permits system 10 to be used as a precise rate sensor.

The construction of gimbal 18, using two concentric cylinders 48, 50 that are welded together, provides a significant simplification in gimbal design than has heretofore been known. Gimbal 18 of the present invention is a particular improvement over conventional gimbals with flexure pivots used on tuned rotor gyros that involve using many individual parts or other flexure concepts that are not suitable for large angular deflections. The present gimbal design consists of only two parts and, using crossed flexures 42, makes it well suited for relatively large angular deflections ±10 deg. Also, using two concentric cylinders 48, 50 and a wire EDM process to machine flexures 42, and optionally deflection stops 72, can be very cost effective compared to other gimbal designs, and it reduces the effects of manufacturing errors on the gimbal performance. The effects of errors are reduced due to the process of using wire EDM to machine one pair of flexures at the same time and then making the 180 deg rotation of one cylinder relative to the other which tends to minimize any adverse effects due to manufacturing errors.

The gimbal design also permits setting the tuned speed of system 10 to a particular value for each customer in a simple and cost-effective way that does require a design modification. This is required because a spacecraft momentum actuator must be able to perform over a relatively large speed range where the particular speed is a mission specific (i.e., customer specific) requirement. Hence each customer will have different tuned speed requirements. The tuned speed of gimbal 18 can also be set by simply reducing the gimbal ring height from a baseline value which is designed for the lowest permissible tuned speed. This does not alter any other design feature in system 10, since ring 18 is free to flutter and the material removed is off the free ends of the ring18 ensuring that the gimbal's interface to shaft 22 and rotor 14 remain unaltered. This process can easily be accommodated for each customer as trimming the gimbal ring height requires only a few extra machining operations prior to welding gimbal 18.

To operate over a relatively large tilt angle range, a tilt sensor is required that can accommodate this range and yet maintain high precision through-out the range. Tilt sensor 24 which uses a reliefed triangular pattern 32 on the ferromagnetic rotor surface and inductive pick-off 34 achieves this requirement and is very cost-effective to implement. Inductive pick-offs used in tuned rotor gyros and other concepts using optical techniques would not function over this required tilt range ±10deg.

The ability of system 10 to measure rates when the device is not operating at a tuned speed and when it is at non-zero angles was previously unachievable. This ability permits system 10 to be used as an actuator at the same time as it is sensing rates. The conventional

theory on tuned rotor gyros only addresses operating at zero angles while at the tuned speed. Therefore, the control approach and the rate sensing algorithms at untuned speeds provide key advantages.

Achieving a nearly constant torque scale factor over the entire tilt range ±10deg. is an innovative feature permits significant improvement of the calibration of the system 10 over the full tilt range. By selecting the proper length of the axially magnetized permanent magnet rings 30 in rotor 14, the resultant rotor torque applied to rotor 14 from the current in torque coils 28 remains nearly constant with tilt angle. Equivalently, if radially magnetized rings are used, then the proper axial spacing of the rings is determinant

Measuring rotor speed from tilt sensors 24 for the motor speed control gives improved precision of the rate sensing, since it reduces the nonlinear precession torques that are imparted to rotor 14 when it is at a non-zero tilt angle. Inductive tilt sensors 24 are also well suited to providing the signals required for rotor spin speed measurements. The traditional approach is to use the drive shaft speed using, for example, hall sensors that are integral to spin motor 26. However, such a high bandwidth control system could cause relatively large precession torques in system 10 when rotor 14 is at large tilt angles (e.g., a few degrees).

The integrated launch restraint system 38 ensures that flexures 42 cannot be over-stressed under launch and handling loads without introducing other mechanisms that introduce new failure modes into system 10. Launch restraint system 38 is completely passive and does not engage unless a large load is applied to rotor 14.

The mirrored surface of the top of rotor 14 permits easy calibration. An auto-collimating telescope can be mounted to a one axis tilt table on which system 10 is also mounted to precisely align rotor 14 at specific angles relative to drive shaft 22. This set-up can be used to statically balance rotor 14 while it is not spinning, and it can also be used to calibrate tilt sensors 24 when rotor 14 is spinning.

High accuracy 3-axis pointing is possible using system 10 and only one additional 2-axis earth sensor (i.e., a yaw sensor is not required) is required to provide a highly accurate and lightweight attitude control system 150. The accuracy of this attitude control system 150 is driven by the earth sensor accuracy and the rate sensing accuracy of system 10, and is not dependent on an environmental disturbance torque requirement. Other approaches using a single bias momentum wheel have an accuracy that is dependent on the magnitudes of the wheel momentum and the environmental torques. Attitude control system 150, based on system 10.

results in a simple 3-axis attitude control system 150 consisting of only two primary components which significantly reduces the mass, power, and cost of attitude control system 150.

For earth pointing missions, the control law uses the measured roll angle from the earth sensor and the rate measurements from system 10 about the roll and yaw axes directly without the need for an attitude determination algorithm.

The form of the control law using the WHECON controller 151 with an additional yaw rate feedback loop 152 provides a form for the controller gain matrix that uses only four independent quantities. Since these four quantities have a practical significance, then it is possible to more easily select the set of parameters that give the best control system performance.

It will be apparent to those skilled in the art that the foregoing is by way of example only. Modifications, variations and alterations may be made to the described embodiments without departing from the scope of the invention which is defined solely in the claims.

We claim:

1. A momentum management system (10) for spacecraft attitude control, having:

a rotor (14) for providing control torques to a spacecraft;

a drive (25) for rotating said rotor (14);

a torque generation device (27) for imparting torque to said rotor (14); and

a gimbal assembly (12) for coupling said drive (25) to said rotor (14);

characterized by said gimbal assembly (12) having a spinning gimbal (18) provided with flexure joints (20) to permit said rotor (14) to tilt in two axes relative to a drive shaft (22) of said drive (25), through a range of angles from about 0 degrees to about 7 degrees under the control of said torque generation device (27).

2. A momentum management system (10) according to claim 1, wherein said flexure joint (20) includes a cross flexure pivot (42) formed of two flexures (42a, 42b) perpendicular to one another.

3. A momentum management system (10) according to claim 2, wherein said gimbal assembly (12) further includes a launch restraint system (38) to limit deflection of said rotor (14).

4. A momentum management system (10) according to claim 3, wherein said launch restraint system (38) includes deflection stops (72) limiting deflection of said flexures (42a, 42b).

5. A momentum management system (10) according to claim 1, wherein the torque generation devices (27) include permanent magnets (30), mounted inside said rotor (14), and torque coils (28) for providing an inductive field to said permanent magnets (30).

6. A momentum management system (10) according to claim 1, further including sensors (24) for measuring tilt and speed of said rotor (14) at non-tuned speeds and non-zero angles.

7. A momentum management system (10) according to claim 6, wherein a triangular pattern (32) is provided on an outer surface of said rotor (14) for sensing by said sensors (24).

8. A gimbal assembly (12), for a momentum management system (10), having a gimbal ring (18) and flexure joints (20) for coupling a drive shaft (22) to a rotor (14), characterized by:

said flexure joints (20) for permitting said rotor (14) to tilt in two axes relative to said drive shaft (22), through a range of angles from about 0 degrees to about 7 degrees under the control of a torque generation device (27).

9. A gimbal assembly (12) according to claim 8, wherein said flexure joint (20) includes a cross flexure pivot (42) formed of two flexures (42a, 42b) perpendicular to one another.

10. A gimbal assembly (12) according to claim 9, wherein said gimbal assembly (12) further includes a launch restraint system (38) to limit deflection of said rotor (14).

11. A gimbal assembly (12) according to claim 11, wherein said launch restraint system (38) includes deflection stops (72) limiting deflection of said flexures (42a, 42b).

12. A gimbal assembly (12) according to claim 8, wherein said gimbal ring (18) includes an outer ring (48) and an inner ring (50).

13. A method of manufacturing a gimbal assembly (12), characterized by the steps of:

(i) machining an inner ring (50) for insertion in an outer ring (48);

(ii) mounting said inner and outer rings (48, 50) on a jig (56);

(iii) machining flexures (42a, 42b) through lateral sides of said inner and outer rings (48, 50);

(iv) inverting one of said inner and outer rings (48, 50) to align said flexures (42a, 42b) perpendicular to one another;

(v) welding said inner and outer rings (48, 50) to form a gimbal ring (18).

14. A method of manufacturing a gimbal assembly according to claim 13, wherein said step of machining uses wire electric discharge machining.

15. A method of manufacturing a gimbal assembly according to claim 13, further including a step of determining a height of said gimbal ring (18) to provide a predetermined tuned speed of a momentum management system (10).

16. A spacecraft attitude control system (150), having

a momentum management system (10);

a control system (90) for providing control signals to said momentum management system (10);

characterized by said momentum management system (10) having a rotor (14) for providing control torques to a spacecraft;

a drive (25) for rotating said rotor (14);

a torque generation device (27) for imparting torque to said rotor (14); and

a gimbal assembly (12) for coupling said drive (25) to said rotor (14), said gimbal assembly (12) having a spinning gimbal (18) provided with flexure joints (20) to permit said rotor (14) to tilt in two axes relative to a drive shaft (22) of said drive (25), through a range of angles from about 0 degrees to about 7 degrees under the control of said torque generation device (27).

17. A spacecraft attitude control system (150) according to claim 16, wherein said control system (90) includes a WHECON controller (151) and a rate feedback controller (150).

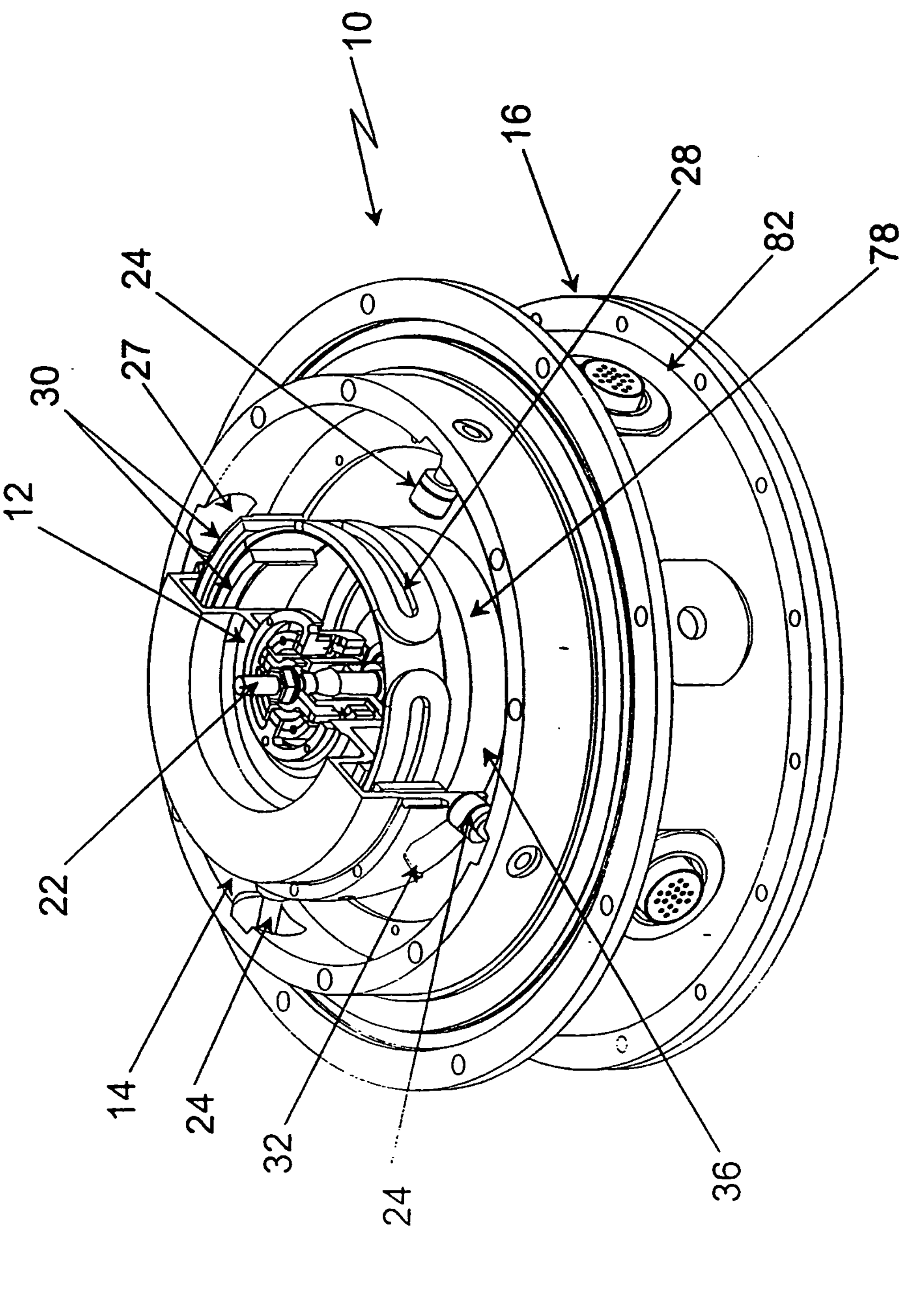

FIG. 1

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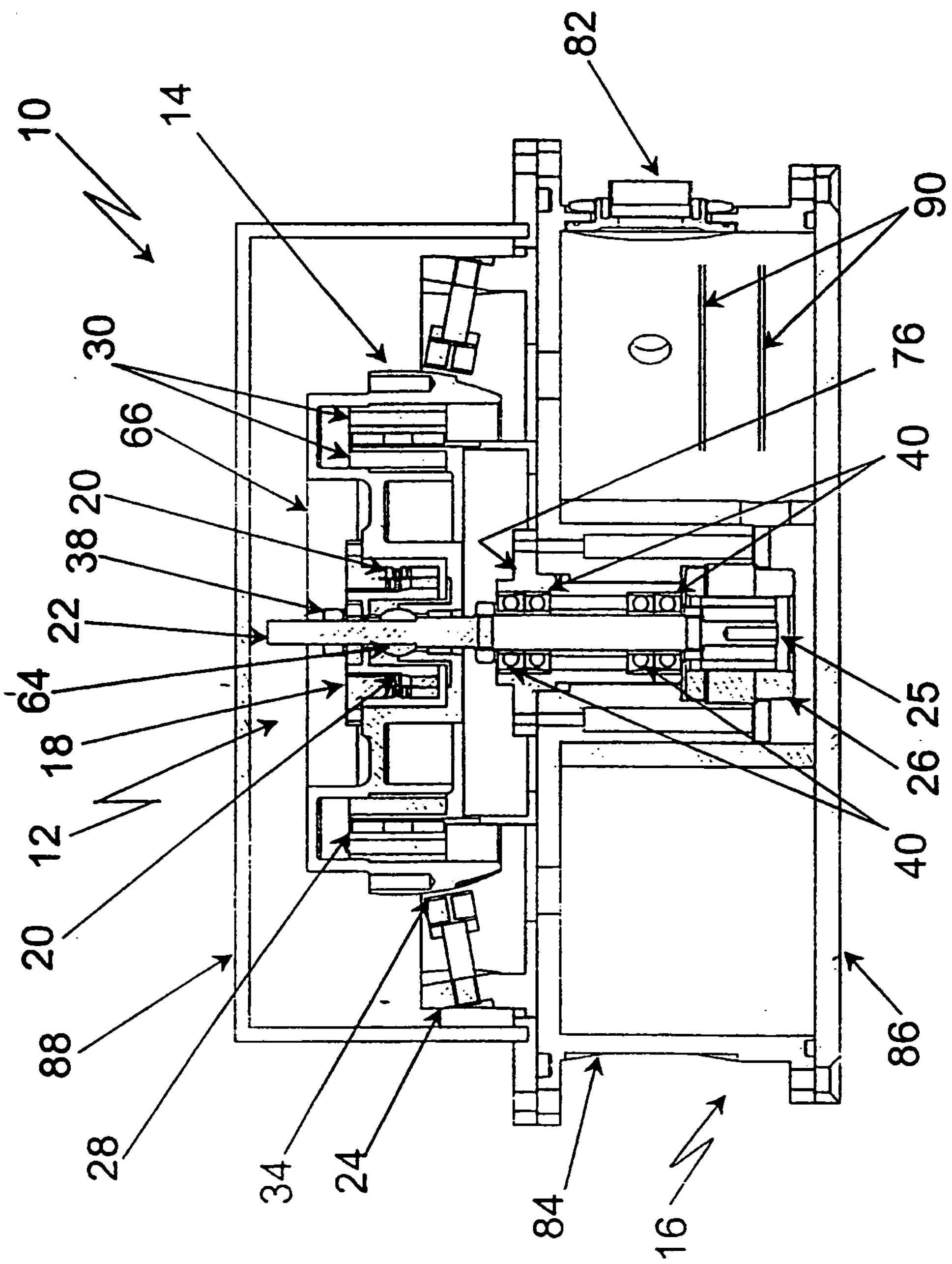

Fig. 2

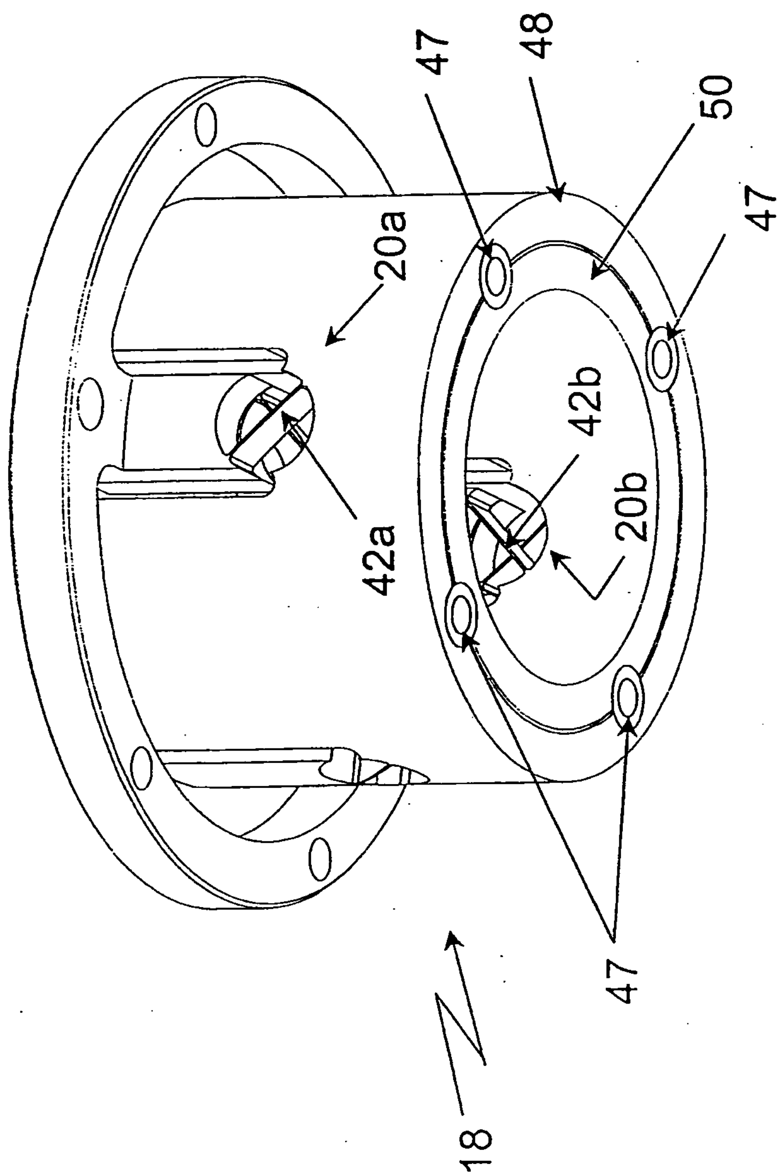

Fig.3

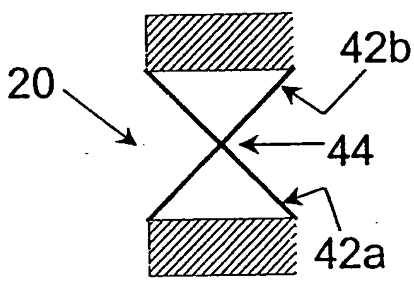

Fig. 4a

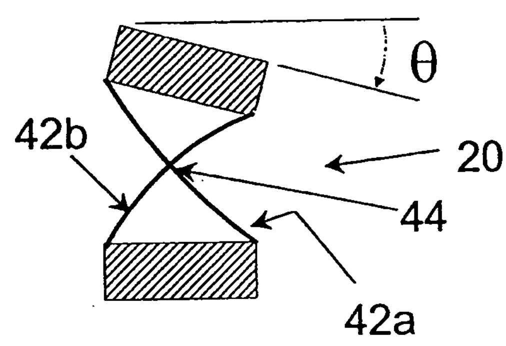

Fig. 4b

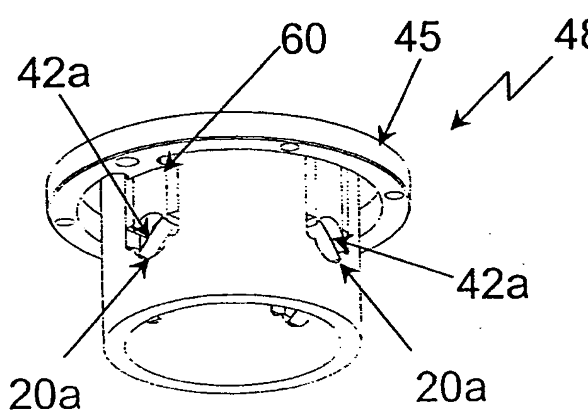

Fig.5a

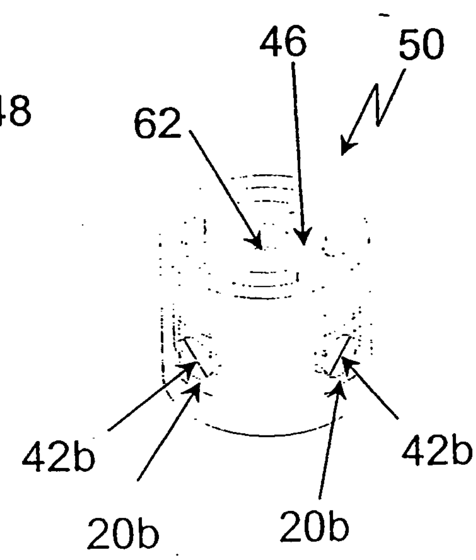

Fig. 5b

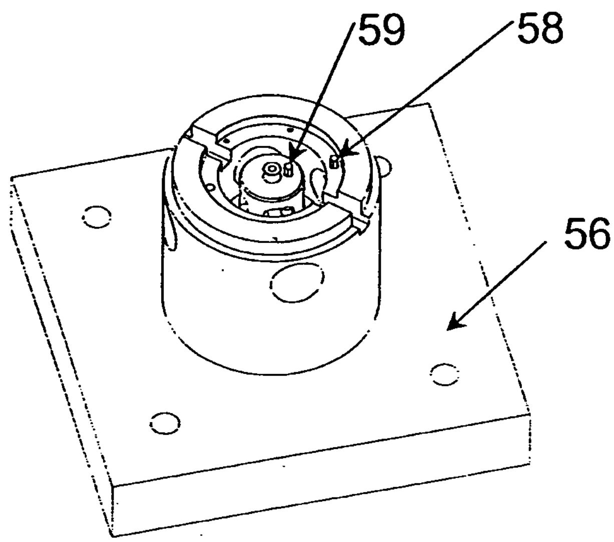
59
58
56

Fig. 6

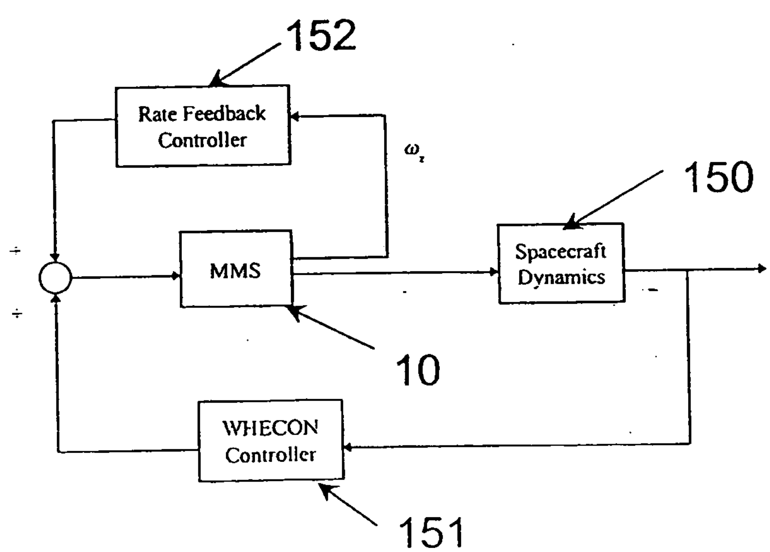
152
Rate Feedback Controller
ω_z
150
Spacecraft Dynamics
MMS
10
WHECON Controller
151

Fig.13

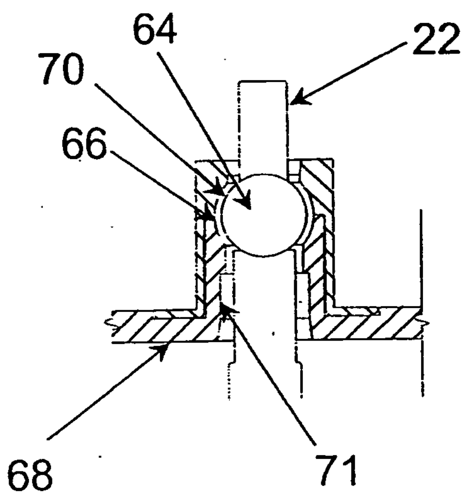

Fig. 7a

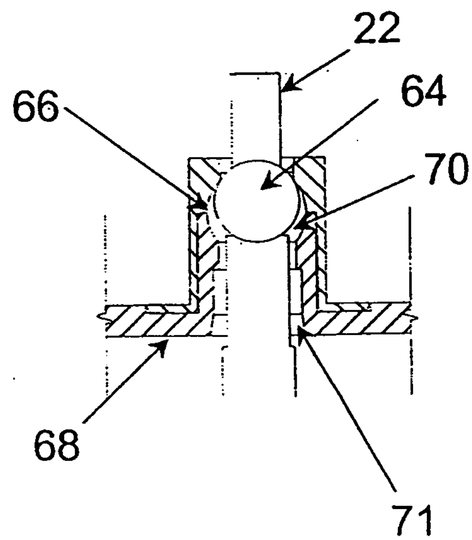

Fig. 7b

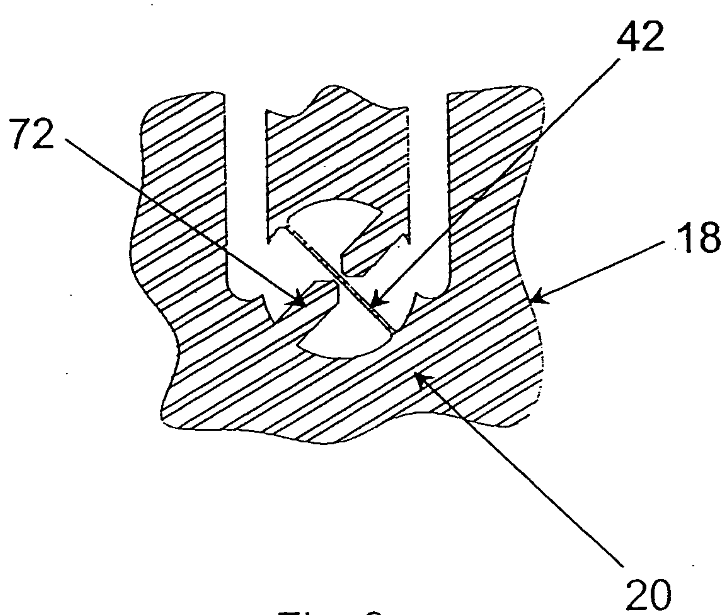

Fig. 8

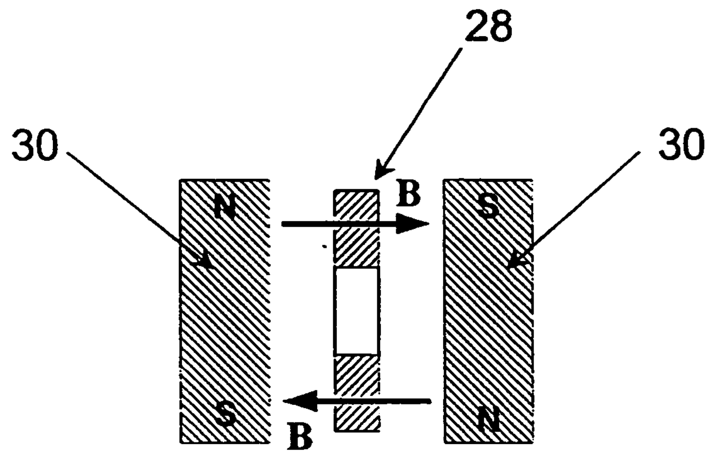

Fig. 9

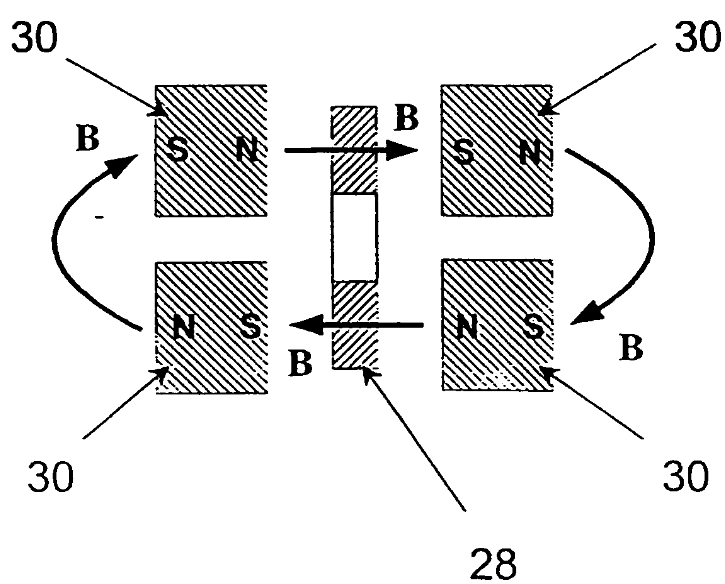

Fig. 10

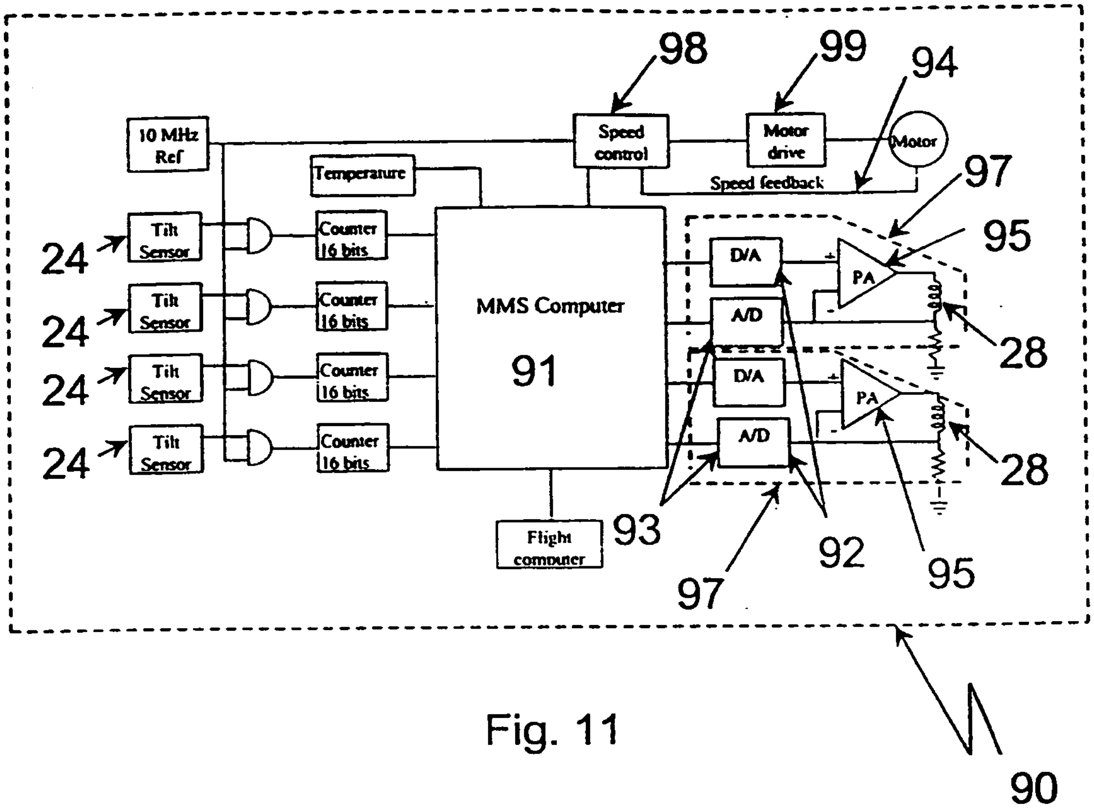
98
99
94
10 MHz Ref
Speed control
Motor drive
Motor
Temperature
Speed feedback
97
95
24
Tilt Sensor
Counter 16 bits
MMS Computer
91
D/A
A/D
PA
28
Flight computer
93
92
97
95
90

Fig. 11

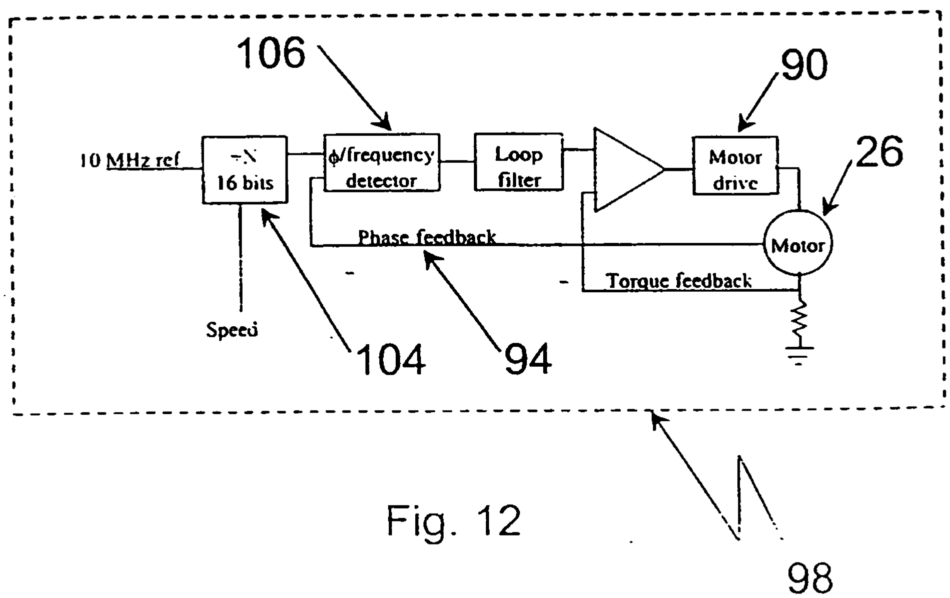
106
90
26
10 MHz ref
÷N 16 bits
ϕ/frequency detector
Loop filter
Motor drive
Motor
Phase feedback
Torque feedback
Speed
104
94
98

Fig. 12

INTERNATIONAL SEARCH REPORT

International Application No
PCT/CA 99/00678

A. CLASSIFICATION OF SUBJECT MATTER
IPC 7 G01C19/22 B64G1/38 F16C11/12

According to International Patent Classification (IPC) or to both national classification and IPC

B. FIELDS SEARCHED

Minimum documentation searched (classification system followed by classification symbols)
IPC 7 G01C B64G F16C

Documentation searched other than minimum documentation to the extent that such documents are included in the fields searched

Electronic data base consulted during the international search (name of data base and, where practical, search terms used)

C. DOCUMENTS CONSIDERED TO BE RELEVANT

Category °	Citation of document, with indication, where appropriate, of the relevant passages	Relevant to claim No.
X	US 4 825 713 A (WILKEY WILLIAM P) 2 May 1989 (1989-05-02) abstract figures 1-3,7-10 column 1, paragraph 1 column 1, line 30 - line 41 column 2, line 1 - line 39 column 2, line 63 -column 4, line 3 ---	1-4,8-16
X	US 4 528 864 A (CRAIG ROBERT J G) 16 July 1985 (1985-07-16) abstract figures 1,2,7-9,13-16 column 1, line 13 - line 47 column 2, line 1 - line 13 column 12, line 48 -column 14, line 41 --- -/--	1-6, 8-11,16

[X] Further documents are listed in the continuation of box C.

[X] Patent family members are listed in annex.

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"P" document published prior to the international filing date but later than the priority date claimed

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"Y" document of particular relevance; the claimed invention cannot be considered to involve an inventive step when the document is combined with one or more other such documents, such combination being obvious to a person skilled in the art.

"&" document member of the same patent family

Date of the actual completion of the international search	Date of mailing of the international search report
30 September 1999	13/10/1999

Name and mailing address of the ISA	Authorized officer
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C.(Continuation) DOCUMENTS CONSIDERED TO BE RELEVANT

Category *	Citation of document, with indication,where appropriate, of the relevant passages	Relevant to claim No.
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INTERNATIONAL SEARCH REPORT

Information on patent family members

International Application No
PCT/CA 99/00678

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